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TECHNICAL NOTE 4006

INVESTIGATION AT TRANSONIC SPEEDS OF DEFLECTORS AND  
SPOILERS AS GUST ALLEVIATORS ON A 35° SWEEP WING

TRANSONIC-BUMP METHOD

By Delwin R. Croom and Jarrett K. Huffman

Langley Aeronautical Laboratory  
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## SUMMARY

An investigation was made in the Langley high-speed 7- by 10-foot tunnel by means of the transonic-bump method to determine the gust-alleviation capabilities of spoilers and deflectors when mounted on a  $35^\circ$  swept semispan wing having NACA 65A006 airfoil sections. The Mach number range was from 0.40 to 1.15, and the angle-of-attack range was from  $-8^\circ$  to or beyond the stall.

The gust-alleviation capabilities (as indicated by the reduction in lift-curve slope) were almost constant (approximately a 20-percent reduction in lift-curve slope) throughout the Mach number range from 0.40 to 1.15 for both the deflector and the spoiler-deflector combination. Increased drag resulted from the deflection of these controls and indicated that they would also be effective as aerodynamic brakes during slowdown to rough-air speed.

At low subsonic speeds the wing with the deflector or the spoiler-deflector control caused no marked effect on the stability of the model; however, at high subsonic speeds (Mach number above about 0.8) the controls caused a decrease in stability and a pitch-up was noted at an angle of attack of about  $6^\circ$  where the lift curve became nonlinear. At supersonic speeds the wings with the controls were less stable than the plain wing; and both wings exhibited pitch-up, as did the plain wing, at an angle of attack of about  $12^\circ$  where the lift curve became nonlinear.

## INTRODUCTION

A previous investigation has shown that spoilers and deflectors when mounted well forward on an unswept-wing airplane model would reduce the normal acceleration due to gusts. (See ref. 1.) As has been pointed out in reference 1, the reduction in normal acceleration is directly proportional to the reduction in lift-curve slope. The investigation at low

speeds of spoilers and deflectors as gust alleviators was extended to include a wing having  $35^\circ$  of sweep and also a 1/4-scale model of the Bell X-5 airplane having  $35^\circ$  swept wings and is reported in reference 2. From that investigation it was found that, in order for a deflector to have the same effectiveness on a sweptback wing as on an unswept wing, the control would have to be located in a more rearward position and would possibly require a larger projection. Results obtained from reference 2 indicate that a deflector extending from the 41-percent- to the 59-percent-semispan station along the 41-percent-chord line (which corresponds to the 38-percent-chord line as defined in ref. 2) when projected 15 percent of the chord would give about 20-percent reduction in lift-curve slope.

The purpose of the present investigation is to determine the lift-curve-slope reduction capabilities of a deflector control and a spoiler-deflector control on a  $35^\circ$  swept semispan wing at high subsonic and transonic speeds.

#### SYMBOLS AND COEFFICIENTS

The forces and moments measured on the model are presented with respect to an orthogonal system of axes. The longitudinal axis is parallel to the free airstream, and the lateral axis is in the wing chord plane. The origin of the axes is at the intersection of the root chord and a line that is perpendicular to the root chord and passes through the quarter-chord point of the mean aerodynamic chord.

b	twice wing span of semispan model, 1.0 ft
c	wing chord, ft
$\bar{c}$	mean aerodynamic chord of wing, 0.255 ft
$c_{av}$	average wing chord, ft
$C_D$	drag coefficient, $\frac{\text{Twice semispan drag}}{qS}$
$C_L$	lift coefficient, $\frac{\text{Twice semispan lift}}{qS}$
$C_{L_\alpha}$	lift-curve slope
$C_m$	pitching-moment coefficient, $\frac{\text{Twice semispan pitching moment}}{qS\bar{c}}$

$\Delta C_m$	incremental pitching-moment coefficient
M	Mach number
q	dynamic pressure, $\frac{\rho V^2}{2}$ , lb/sq ft
R	Reynolds number, based on $\bar{c}$
S	twice wing area of semispan model, 0.250 sq ft
V	free-stream air velocity, ft/sec
$\alpha$	angle of attack, deg
$\rho$	mass density of air, slugs/cu ft

#### MODEL AND APPARATUS

The steel semispan wing model had an angle of sweep of  $35^\circ$  at the quarter-chord line, an aspect ratio 4, a taper ratio of 0.6, and an NACA 65A006 airfoil section parallel to the free airstream. A drawing of the wing with pertinent dimensions and data is shown in figure 1.

The wing was equipped with a deflector and a spoiler-deflector combination. The projection of the deflector was 15 percent of the average local chord and extended from  $0.41b/2$  to  $0.59b/2$  along the 40.7-percent-chord line. The projection of the spoiler was 2.5 percent of the average local chord, was of the same span as the deflector, and extended along the 35.7-percent-chord line on the upper surface.

The model was mounted on an electrical strain-gage balance enclosed within the bump, and the longitudinal aerodynamic forces and moments were recorded by means of calibrated recording potentiometers. The model butt passed through a hole in the turntable in the bump surface. Leakage through this hole was kept to a minimum by the use of a sponge seal fastened to the under surface of the bump turntable.

#### TESTS AND CORRECTIONS

The model was tested in the flow field of a bump mounted on the floor of the Langley high-speed 7- by 10-foot tunnel. The Mach number range was from 0.40 to 1.15 and the angle-of-attack range was from  $-8^\circ$  to or beyond the stall. There is a small Mach number variation over the

wing for a given test Mach number, and charts showing the Mach number gradient over the bump with the model removed are given in reference 3. The variation with Mach number of mean test Reynolds number based on the mean aerodynamic chord is given in figure 2.

No corrections to the data have been applied. The usual wind-tunnel blockage and jet-boundary corrections are considered negligible because of the small size of the model compared with the size of the tunnel test section.

## RESULTS AND DISCUSSION

The lift, drag, and pitching-moment coefficients are presented as functions of angle of attack in figure 3 for the plain-wing, deflector, and spoiler-deflector configurations for Mach numbers from 0.40 to 1.15. A summary plot of the lift-curve slope  $C_{L_\alpha}$  (measured at  $C_L \approx 0.3$ ), the angle of attack at  $C_L = 0.3$ , and the incremental change in pitching-moment coefficient from  $\alpha = 4^\circ$  to  $\alpha = 8^\circ$  is presented as a function of Mach number in figure 4.

The wing with the deflector or the spoiler-deflector control reduced the lift-curve slope about 20 percent (measured at  $C_L \approx 0.3$  which approximates the average slope between  $C_L = 0$  and the nonlinear portion of the lift curve which occurs between an angle of attack of  $6^\circ$  and  $12^\circ$ ). This reduction agrees very well with the lift-curve-slope reduction obtained in the low-speed tests reported in reference 2. Inasmuch as this type of control is effective in reducing the lift-curve slope on a swept-wing model throughout the Mach number range, it should be effective as a gust alleviator throughout the speed range. It should be noted, however, that the wing with the controls exhibited greater nonlinearities in aerodynamic characteristics between an angle of attack of  $6^\circ$  and  $12^\circ$  than did the plain wing configuration. (See fig. 3.)

The attitude change, a change in angle of attack for a given lift coefficient over the linear portion of the lift curve, was very small because of the addition of the controls. (At  $C_L = 0.3$  the maximum change was only  $2^\circ$ .)

Some scatter was noted in the drag data at the lower Mach numbers. (See fig. 3.) However, the drag of the wing was increased by the addition of the deflector or spoiler-deflector controls; thus, this increase in drag indicates that the controls would also be effective as aerodynamic brakes during slowdown to rough-air speed.

At low subsonic speeds the deflector or the spoiler-deflector control had no marked effect on the longitudinal stability of the model; however, at high subsonic speeds the controls caused a decrease in stability and a pitch-up was noted at an angle of attack of about  $6^\circ$  where the lift curves became nonlinear. At supersonic speeds the wings with the control configurations were less stable than the plain wing; and both wings exhibited pitch-up, as did the plain wing, at an angle of attack of about  $12^\circ$  where the lift curve became nonlinear. (See fig. 3.) The change in pitching-moment coefficient from  $\alpha = 4^\circ$  to  $\alpha = 8^\circ$ , as shown in figure 4, indicates that at moderate lift coefficients ( $C_L \approx 0.3$  to  $C_L \approx 0.6$ ) the destabilizing effects due to controls would be largest at a Mach number of about 0.95.

#### CONCLUDING REMARKS

The gust-alleviation capabilities, as indicated by the reduction in lift-curve slope, were almost constant (approximately a 20-percent reduction in lift-curve slope) throughout the Mach number range from 0.40 to 1.15 for both the deflector and the spoiler-deflector combination. Increased drag resulted from the deflection of these controls and indicated that they would also be effective as aerodynamic brakes during slowdown to rough-air speed.

At low subsonic speeds the wing with the deflector or the spoiler-deflector control caused no marked effect on the stability of the model; however, at high subsonic speeds (Mach number above about 0.8) the controls caused a decrease in stability and a pitch-up was noted at an angle of attack of about  $6^\circ$  where the lift curve became nonlinear. At supersonic speeds the wings with the controls were less stable than the plain wing; and both wings exhibited pitch-up, as did the plain wing, at an angle of attack of about  $12^\circ$  where the lift curve became nonlinear.

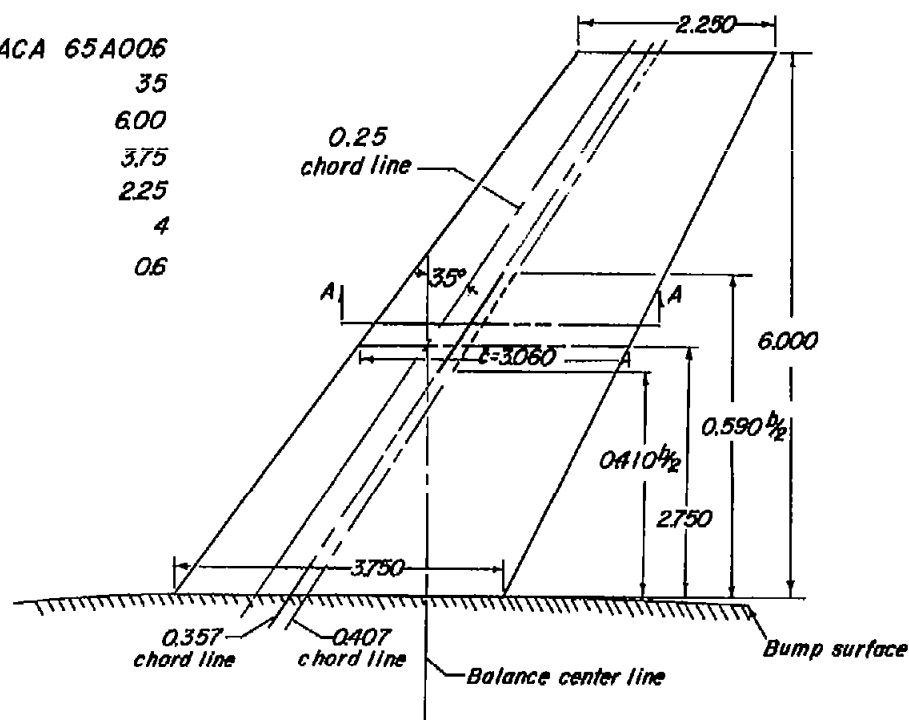
Langley Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., March 27, 1957.

## REFERENCES

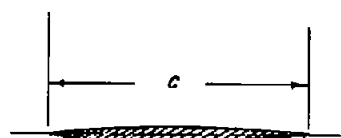
1. Croom, Delwin R., Shufflerbarger, C. C., and Huffman, Jarrett K.: An Investigation of Forward-Located Fixed Spoilers and Deflectors as Gust Alleviators on an Unswept-Wing Model. NACA TN 3705, 1956.
2. Croom, Delwin R., and Huffman, Jarrett K.: Investigation at Low Speeds of Deflectors and Spoilers as Gust Alleviators on a Model of the Bell X-5 Airplane With  $35^\circ$  Swept Wings and on a High-Aspect-Ratio  $35^\circ$  Swept-Wing—Fuselage Model. NACA TN 4005, 1957.
3. Croom, Delwin R., and Wiley, Harleth G.: Investigation at Transonic Speeds of the Hinge-Moment and Lift-Effectiveness Characteristics of a Single Flap and a Tandem Flap on a  $60^\circ$  Delta Wing. NACA RM L53E28a, 1953.

*General Dimensions*

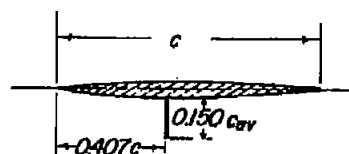
<i>Airfoil section</i>	NACA 65A006
<i>Sweepback of the quarter-chord line, deg</i>	35
<i>Semispan, in.</i>	6.00
<i>Root chord, in.</i>	3.75
<i>Tip chord, in.</i>	2.25
<i>Aspect ratio</i>	4
<i>Taper ratio</i>	0.6



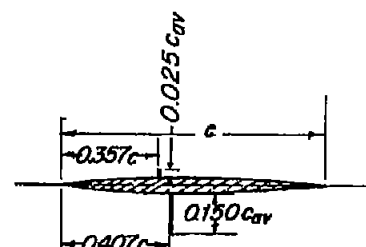
*Section A-A*



*Plain-wing configuration*



*Deflector configuration*



*Spoiler-deflector configuration*

Figure 1.- Sketch of model showing deflector and spoiler-deflector details.

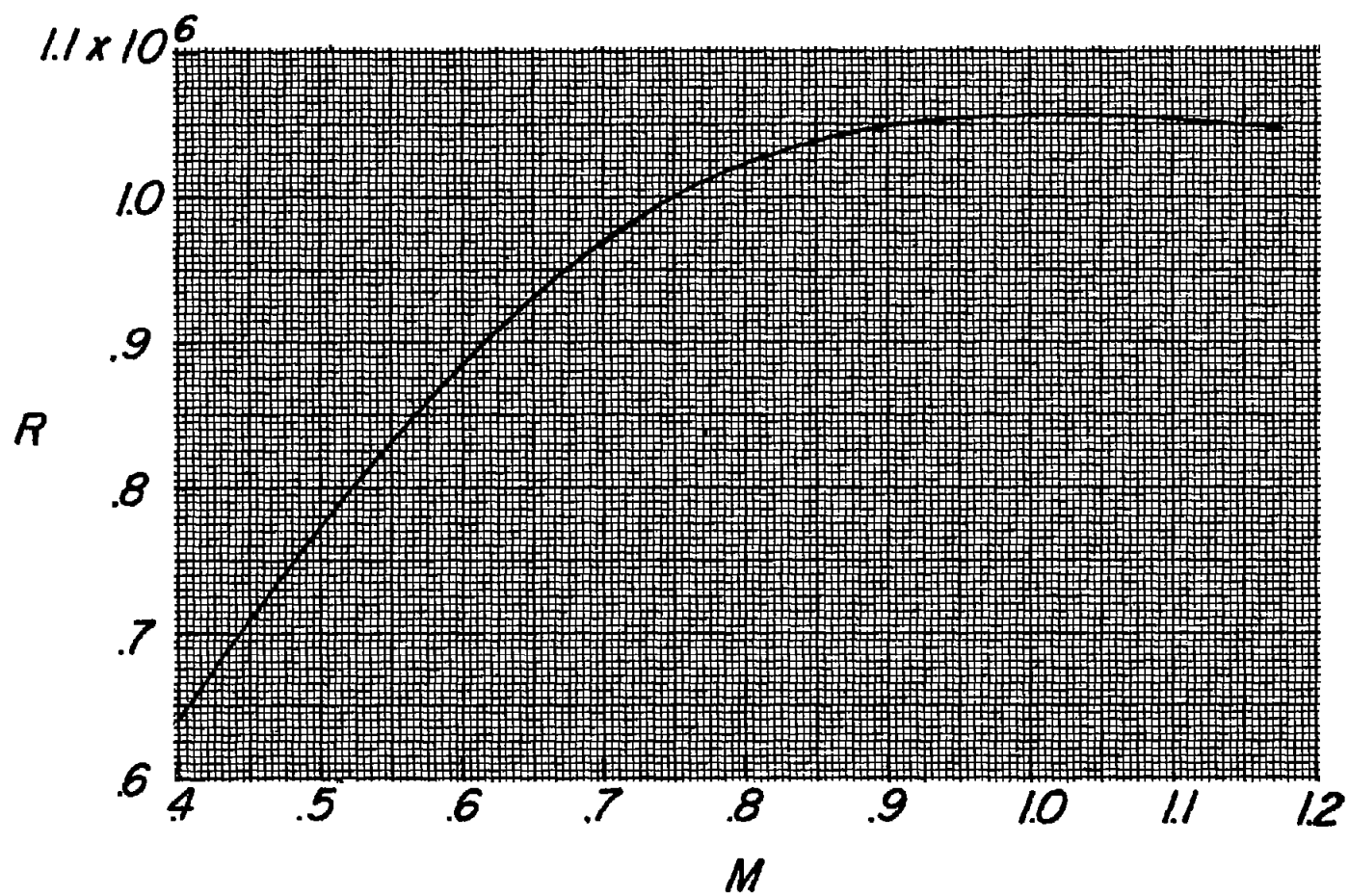
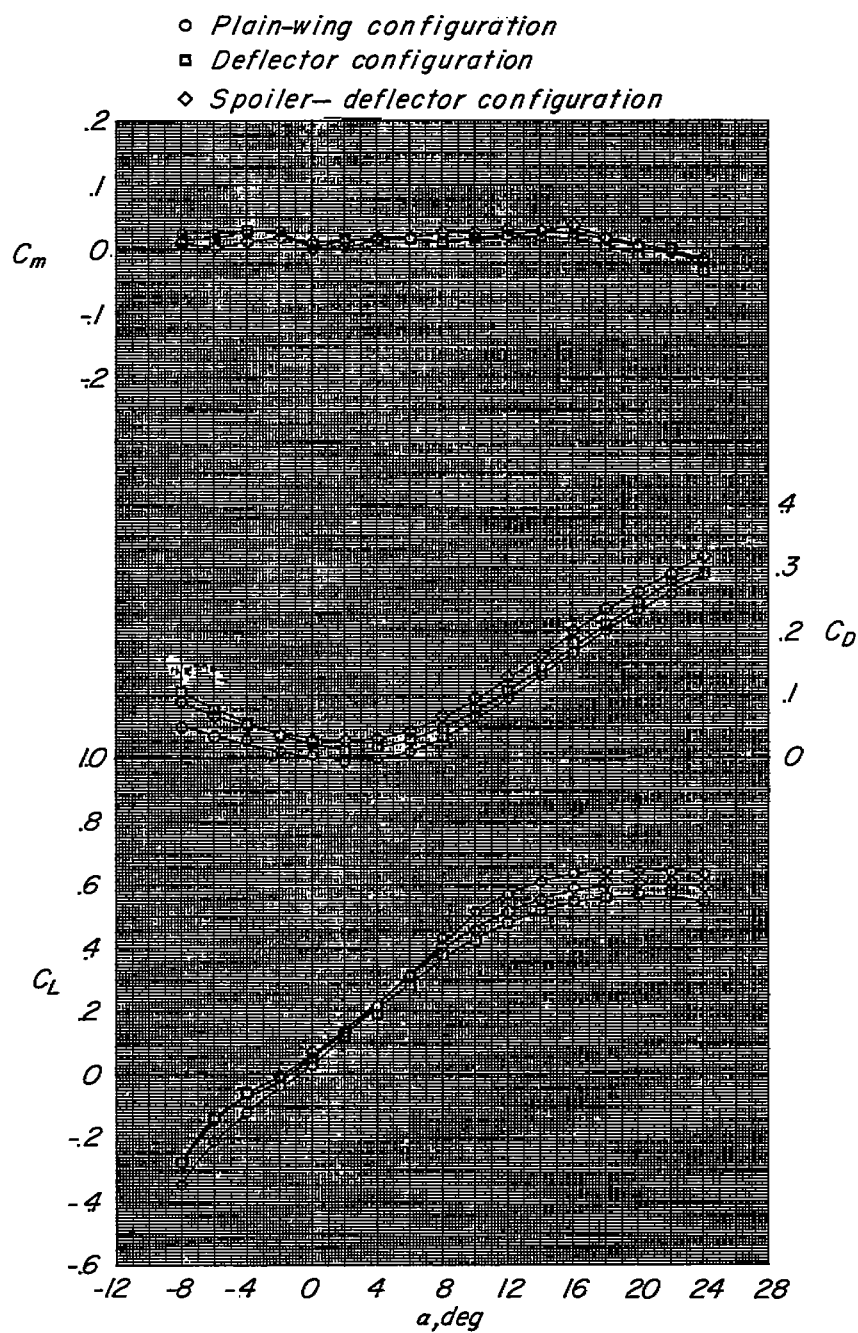
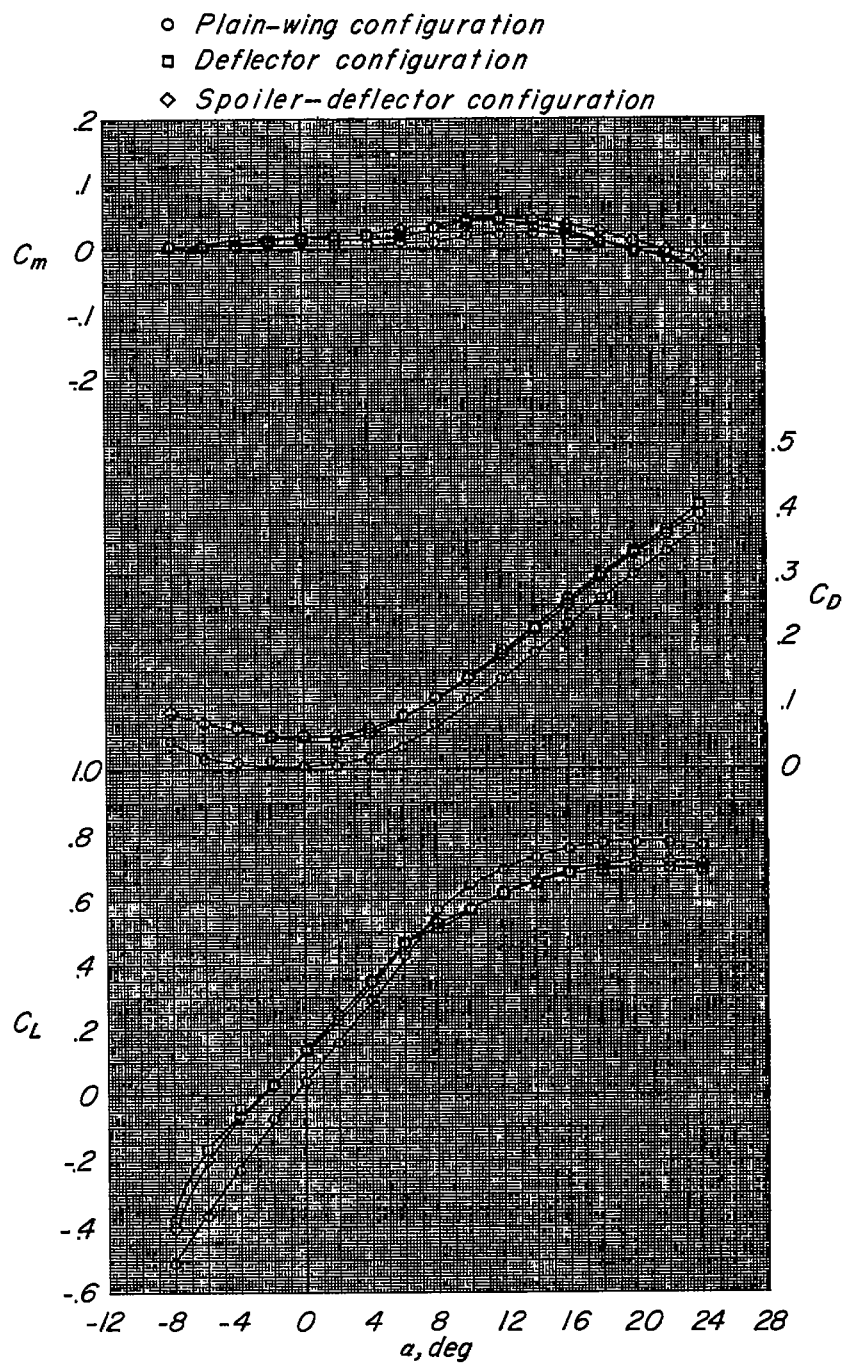


Figure 2.- Variation of mean test Reynolds number with Mach number.



(a)  $M = 0.40$ .

Figure 3.- Aerodynamic characteristics in pitch of plain-wing, deflector, and spoiler-deflector configurations.



(b)  $M = 0.60$ .

Figure 3.- Continued.

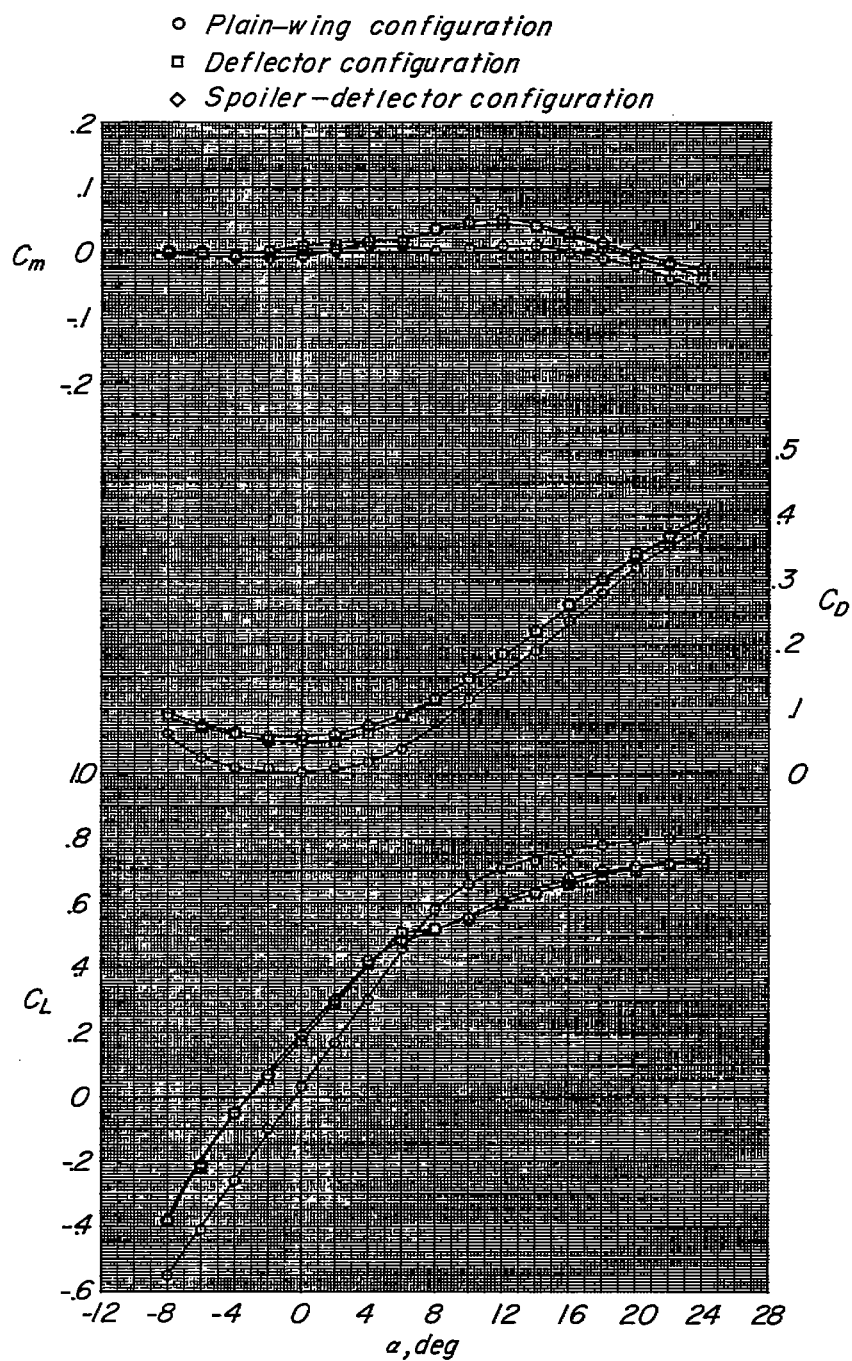
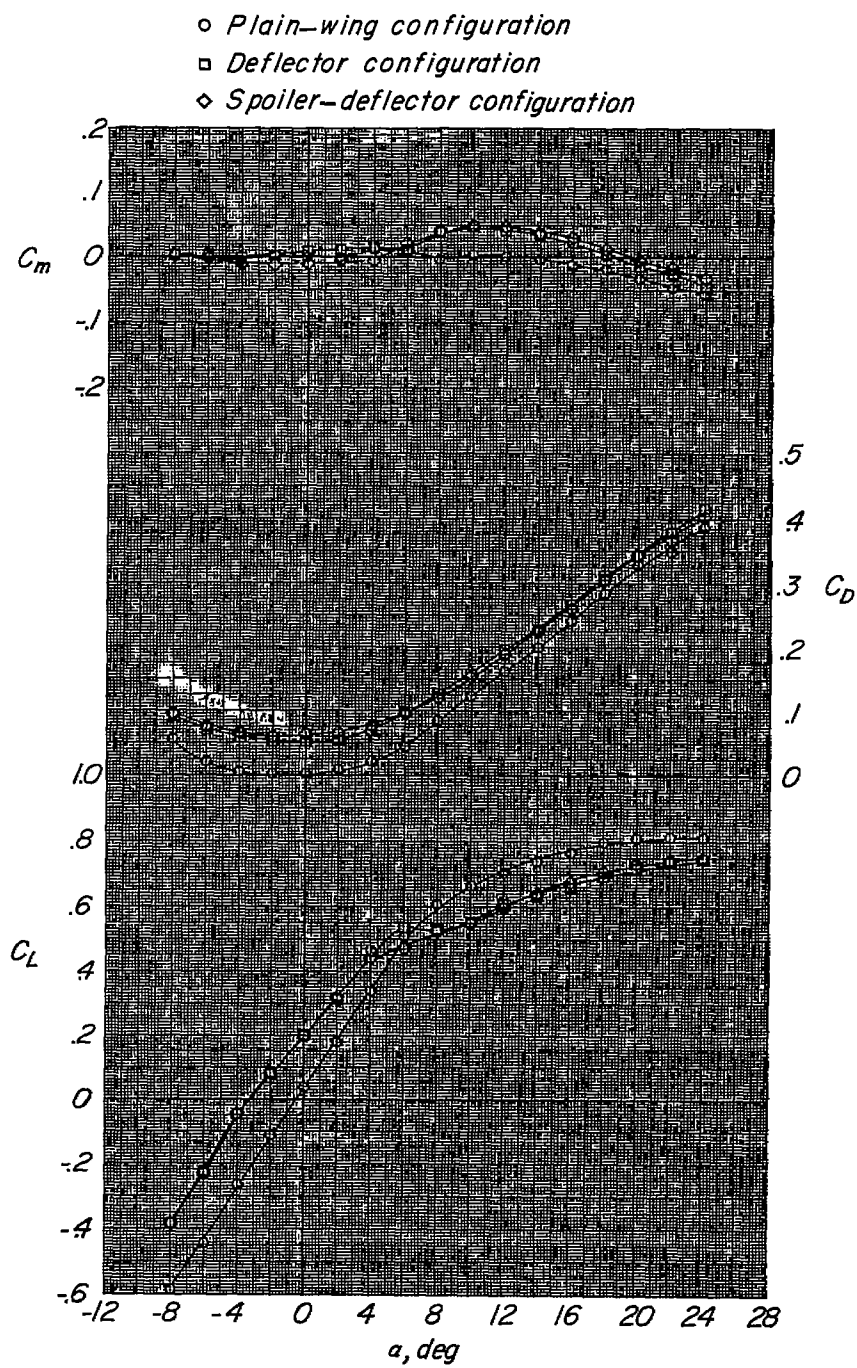
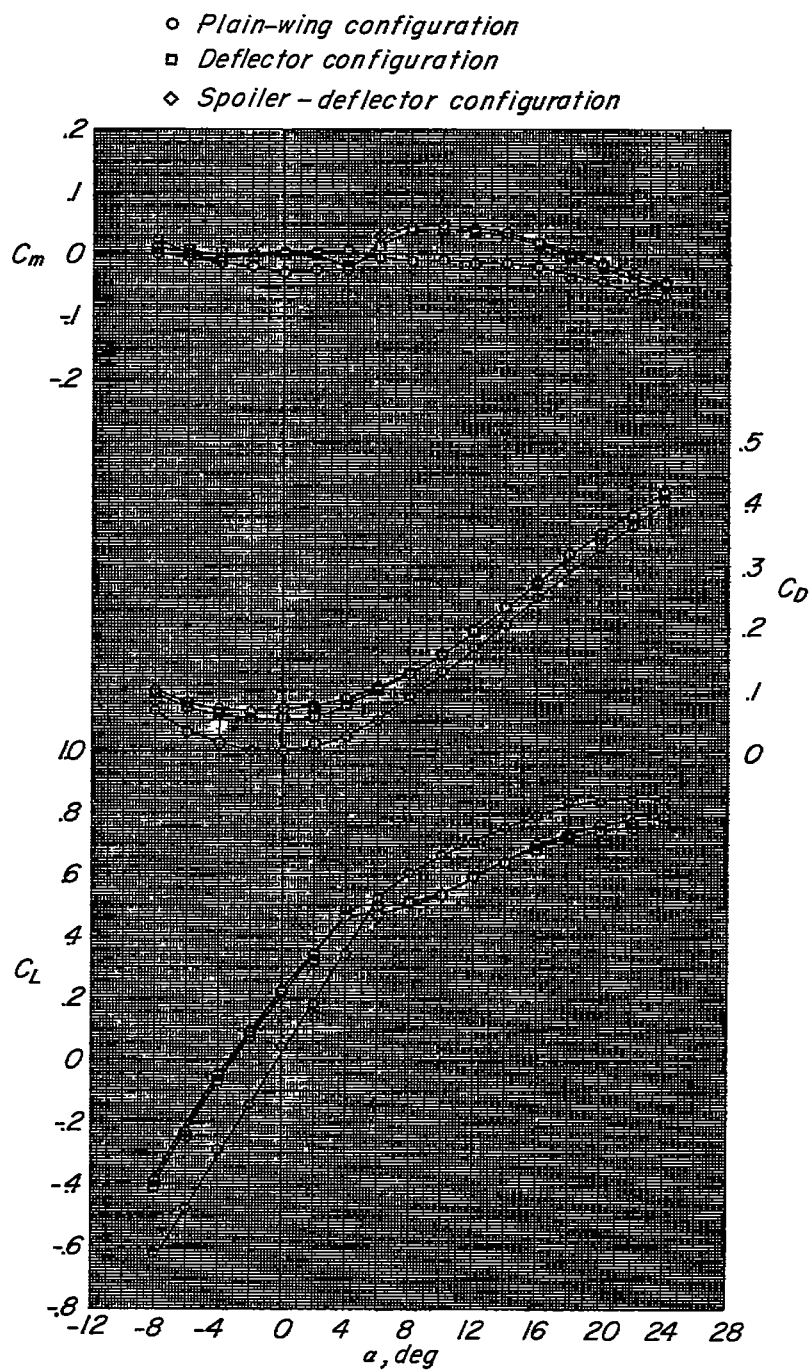
(c)  $M = 0.8$ .

Figure 3.- Continued.



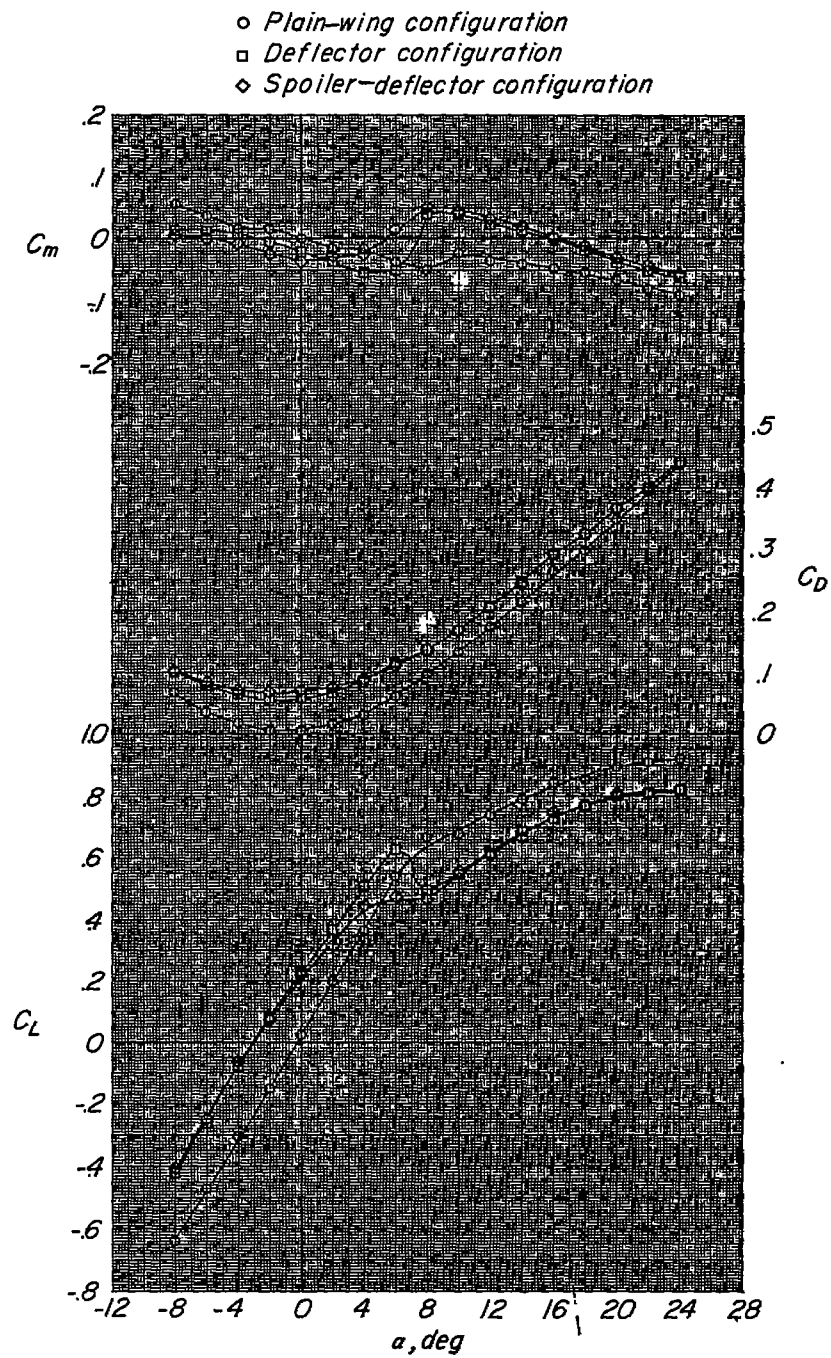
(a)  $M = 0.85$ .

Figure 3.- Continued.



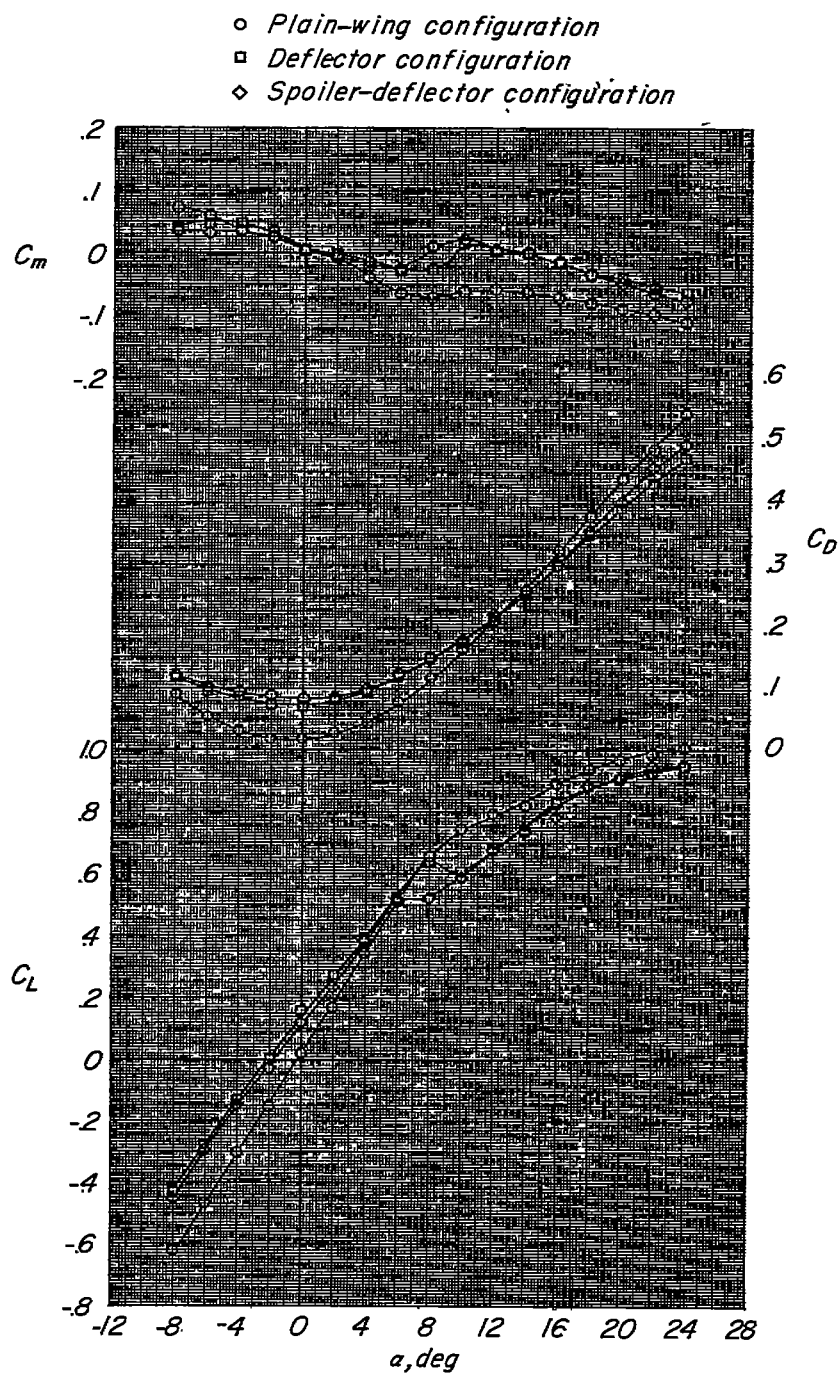
(e)  $M = 0.90$ .

Figure 3.- Continued.



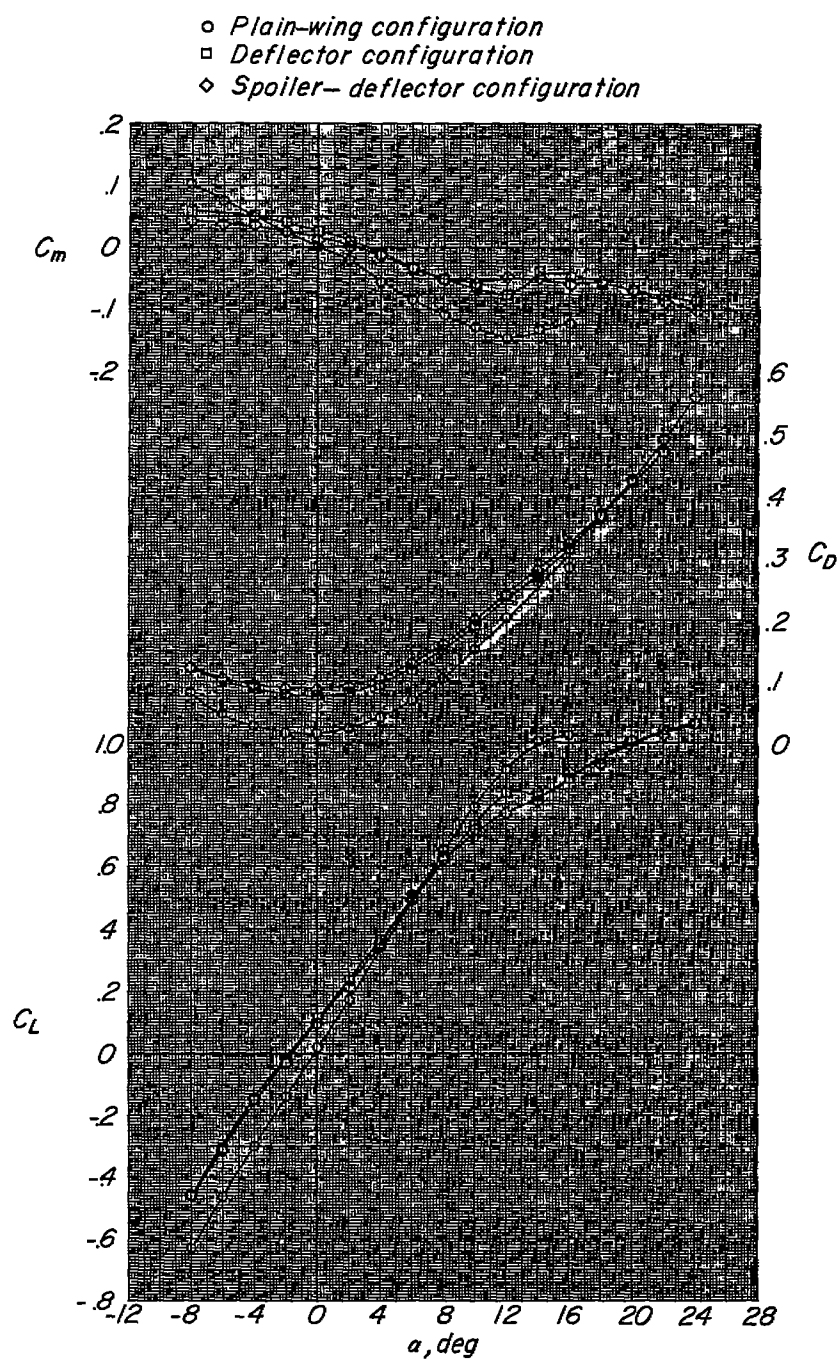
(f)  $M = 0.95$ .

Figure 3.- Continued.



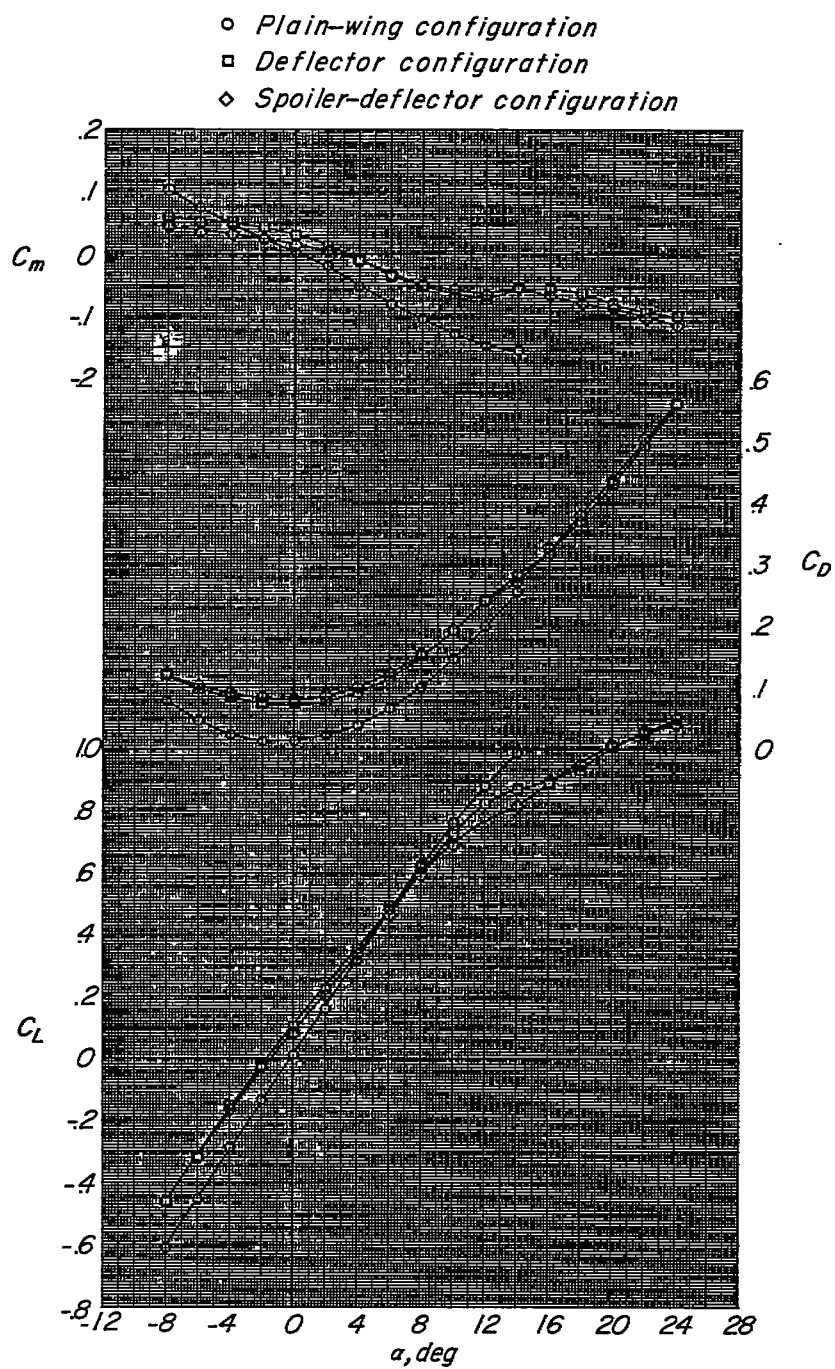
(g)  $M = 1.00$ .

Figure 3.- Continued.



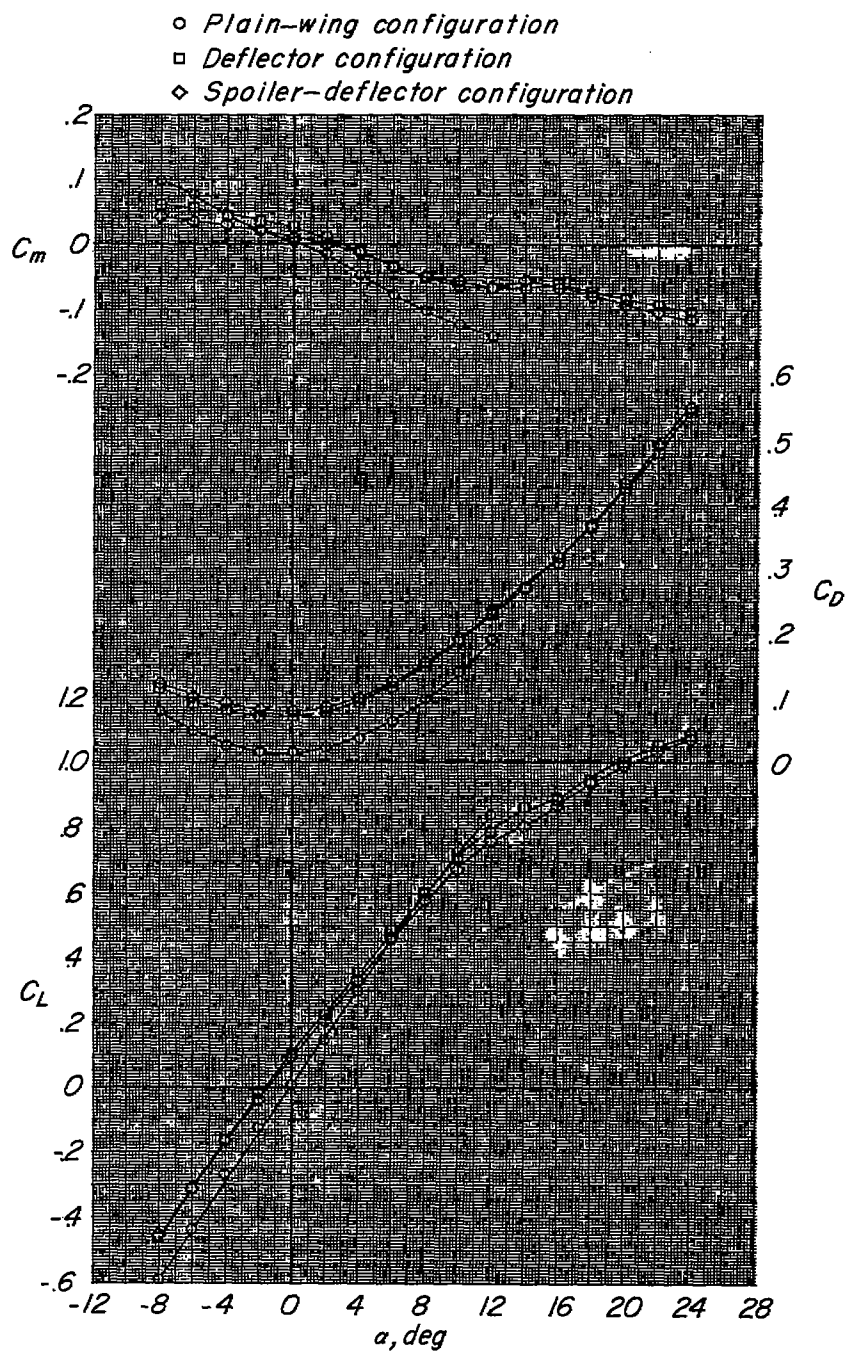
(h)  $M = 1.05$ .

Figure 3.- Continued.



(i)  $M = 1.10$ .

Figure 3.- Continued.



(j)  $M = 1.15$ .

Figure 3.- Concluded.

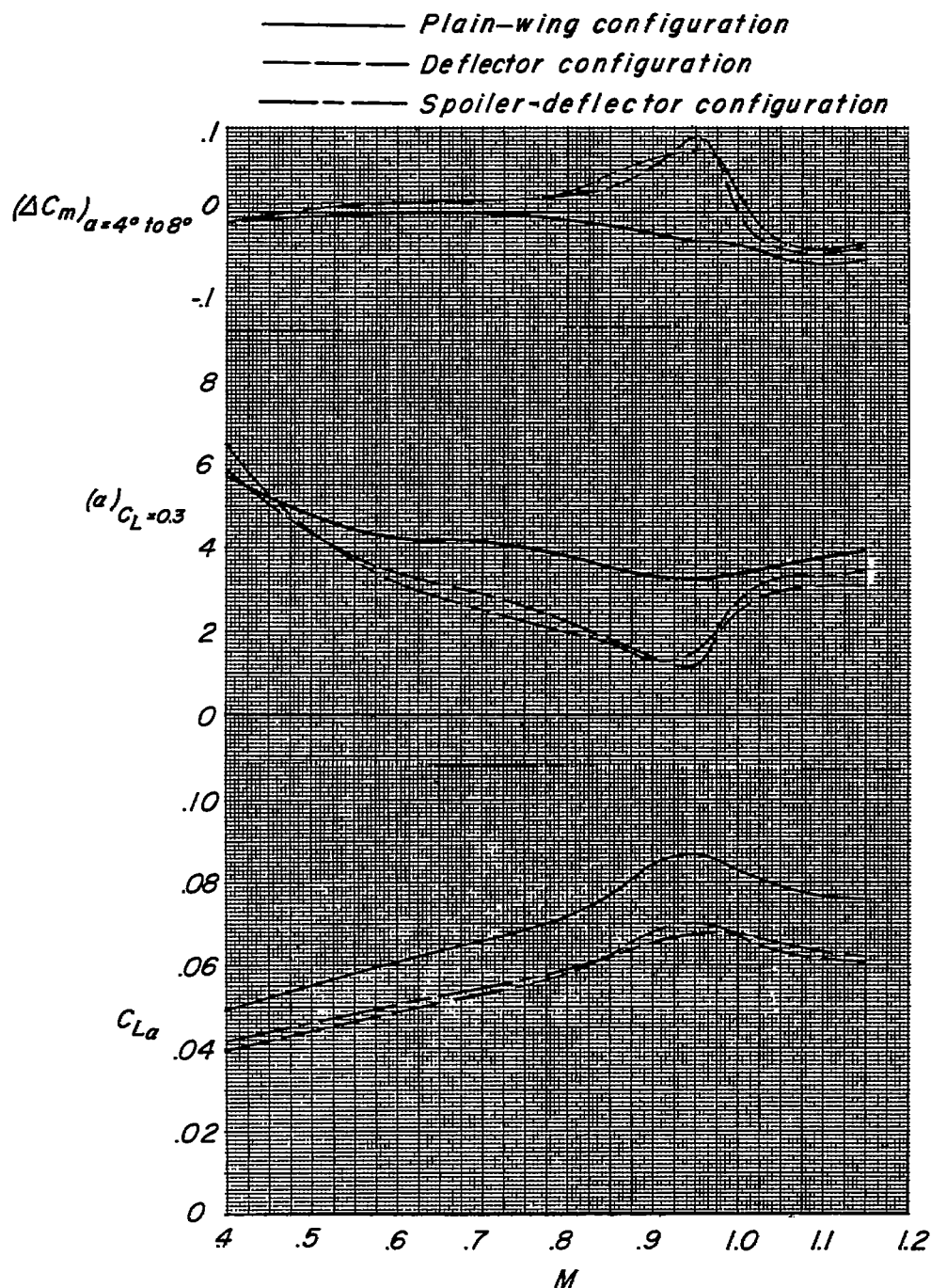


Figure 4.- Variation of lift-curve slope (slope taken at  $C_L \approx 0.3$ ), angle of attack at  $C_L = 0.3$ , and incremental pitching-moment coefficient from  $\alpha = 4^\circ$  to  $\alpha = 8^\circ$  as a function of Mach number.